Thesis
F2

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DESIGN LOADS FOR HORIZONTAL TAIL SURFACES FOR AIRPLANES.

By

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For the Department of
Aeronautical Engineering.

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OBJECT:

The object of this report is to present a simple, rational method for determining the value of design loads to be used for airplane horizontal tail surfaces and fuselage tail structures.

INTRODUCTION:

Investigation establishes the fact that the design loads sought, must be determined from two distinct conditions of flight. The basic assumptions made are:

(1) That the dive condition at terminal velocity, or at some limiting velocity, depending upon the type and service expected of the aircraft, will design the front spar of the horizontal stabilizer.

(2) That the abrupt pull-up condition from some limiting velocity, depending upon the type and service expected of the aircraft, will design the rear spar of the horizontal stabilizer and the elevator.

These assumptions are conclusively borne out by an examination of the pressure distribution tests conducted by the N.A.C.A., upon the tail surfaces of the F6C-4 and PW-9 airplanes.

More specifically then, to arrive at a successful accomplishment of this investigation, the following determinations must be made:

(1) A reasonably simple and rational expression that will give the total normal tail load upon the horizontal surfaces at the particular limiting diving speed specified for the type.

(2) A distribution of the normal tail load found in (1), in such a manner as to impose a critical loading on the front spar of the horizontal stabilizer.

(3) A reasonably simple and rational expression, that will give the total normal tail load upon the horizontal surfaces for the abrupt pull-up condition, at a particular limiting diving speed for the type, from which the abrupt pull-up shall be executed.

(4) A distribution of the total normal tail load found in (3), in such a manner as to impose a critical loading upon the rear spar of the horizontal stabilizer.

A special effort has been made in the compilation of this report, to include all pertinent calculations and references, in order that the results attained may be substantiated by the material to be found within its covers. These data and comparative calculations are contained in the " APPENDIX".

DISCUSSION:

Dive Condition: For the purpose of determining the total normal tail load, calculations were first made upon the F6C-4 airplane by the method set forth in A.D.M. 1061. A limited amount of basic data necessary to the determination of the normal tail load were available. In point, stagger, relative wing efficiency, and equivalent monoplane aspect ratio had to be determined. The calculation of the tail load was, therefore, extended somewhat for this airplane. But granting this extension, the impression received was to the effect that the determination of the normal

tail load by the method of A.D.M. 1061, was exceedingly laborious. And having in mind the specific purpose of this investigation, to wit,- to attain the desired results by simple and rational methods, the investigation naturally proceeded to the study of means by which such simplification could be brought about.

The N.A.C.A. Report No. 307 "Pressure Distribution over Tail Surfaces of F6C-4 Airplane", specified neither the test weight of the airplane , nor the C.G. location. In order to make a precise comparison between calculated tail loads and loads obtained as the result of actual flight tests, the tail load calculations for the F6C-4 airplane were repeated. The gross weight and C.G. location for the first set of calculations, were assumed to be those corresponding to the fully loaded condition. In an indirect manner later information was received , which indicated that an airplane with a gross weight and C.G. location different from that of the fully loaded airplane, had been used in the N.A.C.A. tests, and these values were employed in the second set of calculations on the F6C-4.

It was the comparison of the results obtained in these two sets of calculations, that shed light upon a possible simplification of the tail load formula given in A.D.M. 1061. The results indicated that near'zero lift' for the airplane, the values of $K_{M\ c.g.}$ agreed almost absolutely, for the two C.G. locations.

Comparative calculations of $K_{M\ c.g.}$ were then made for extreme C.G. locations, and the results plotted in Fig. 22. The curves showed that very wide ranges of C.G. location, effected but slight

changes in the value of $K_{M\ c.g.}$ And for the usual range of C.G. location employed in actual design, the change in $K_{M\ c.g.}$ would be much smaller. It was then decided that the airfoil moment coefficient at zero lift, might well be employed in the tail load formula, instead of $K_{M\ c.g.}$, which is determined only after lengthy calculations. The agreement found to exist, by reason of using the value of K_M of the airfoil at zero lift, and of using $K_{M\ c.g.}$ for the airplane, upon the resulting tail load, was pronounced "good".

For the F6C-4 airplane, calculations by the method of A.D.M. 1061, gave a tail load of 925 lbs. at 247 m.p.h., as compared with 468 lbs. at the same speed recorded by actual pressure tests by the N.A.C.A. The calculations give an average tail load of 28 lbs. per sq.ft., while the average tail load by actual flight test is but 14 lbs. per sq.ft. The calculated normal tail load for the dive condition at terminal velocity of 286 m.p.h., would be 1240 lbs., or 38 lbs. per sq.ft. The normal tail load at 286 m.p.h. determined on the basis of actual flight test at 247 m.p.h., would be 630 lbs., or an average loading of but 19 lbs. per sq. ft. Since this load would correspond to a design condition, it appears that either 50% of the calculated normal tail load is borne by the fuselage, or that pressure tests recorded low readings. It can hardly be disputed that 38 lbs. per sq.ft. is a more reasonable value for design load, than is 19 lbs. per sq.ft. for this type of aircraft.

It is unquestionably true that a considerable part of the total balancing load on the tail will be borne by the fuselage. It does not seem likely that as much as 50% would be so borne, however.

The form of the particular fuselage aft of the C.G. would be a governing factor. At this point the conclusion must be drawn, to the effect that the actual normal tail load borne by the horizontal tail area, is at present an indeterminate quantity, and will remain as such, until a vast amount of pressure distribution data, covering both fuselage and tail surfaces on a large number of airplanes, is at hand. The calculated normal tail loads, therefore, will always be approximate and conservative in varying degrees, on various airplanes. Then too, the results obtained cannot possess greater precision than the basic airfoil data, and these are not in close agreement. A.D.M. 1061 gives a terminal velocity of 312 m.p.h. for the PW-9 airplane. For this same airplane, Gottingen tunnel data, which formed the basis for PW-9 computations, by the method of A.D.M.1061, contained in this report, gives a terminal velocity of 330 m.p.h. The maximum normal tail load as computed in A.D.M. 1061 is 1480 lbs.; this difference in terminal velocity alone, would result in an increase in tail load to 1660 lbs. It appears, therefore, that any attempt to calculate with precision, the actual normal tail load upon the horizontal tail surfaces alone, is of academic interest only, and a fruitful source of wasted time and effort. However, if with a reasonable expenditure of time and effort, a value of the design normal tail load can be calculated, which will form a basis for tail surface design, it might possess some virtue. A method for determining the value of the design normal tail load by an approximation to the method indicated in A.D.M. 1061, has been evolved in

this report; and so far as comparative calculations have proceeded, reasonable results have been secured.

The expression for normal tail load set forth in A.D.M. 1061 is as follows:

$$\text{Normal Tail Load} = K_{M\ c.g.} \times A_W \times \frac{C}{d} \times V^2$$

It is proposed that $K_{M\ c.g.}$ be replaced by airfoil K_M at zero lift, and that V be equal to the maximum limiting diving speed for the particular type of aircraft under consideration. For the pursuit or fighting, attack and training types of aircraft, V will be the terminal velocity of the airplane as given from the following formula:

$$V_t = \sqrt{\frac{V^3_{max} \times W}{375 \times \eta_{max} \times B.H.P.}}$$

The origin of this formula is unknown. Values of terminal velocity for various types of aircraft have been computed by the above formula, and also by the drag formula, $W \cos 90^\circ = K_x A_W V_t^2$

Phenomenally close agreement of values of terminal velocity, calculated from both these formulas was secured, and the comparative results have been set forth in the "APPENDIX" to this report on page 75. The employment of the power absorbed formula for terminal velocity, obviates the necessity of first determining airplane K_x.

For the purpose of calculating design normal tail loads, for other than the single seater fighter, attack and training types, limiting velocities have been proposed, and have been set forth in this report in "RESULTS".

The use of the zero lift airfoil moment coefficient, in the formula for normal tail load, with velocities less than terminal, is an approximation; but since at very high speeds, great changes in speed are

accompanied by slight changes in angle of attack, which are accompanied by corresponding small changes in $K_{H\ c.g.}$, this approximation can be justified.

The distribution of the normal tail load was worked out from N.A.C.A. pressure distribution data on the F6C-4 and PW-9 airplanes, and an expression for design load in lbs. per inch of run, for the front spar of the horizontal stabilizer has been evolved. These data will be found in this report under "RESULTS".

Pull-up Condition: The Rhode formula for the pull-up condition has been examined, but since its application in its present state of development, is limited to the fighting type of aircraft, another viewpoint was assumed. It appeared that the most severe loading on the horizontal tail surfaces for the pull-up condition, would be the arithmetic sum of the normal tail load that existed by reason of the dive condition, at the instant movement of elevator was begun, and the normal load imposed on the rear spar, by reason of the inclination of the elevator at an angle of 25°. This assumes no loss of velocity and no movement of the tail to a new angle of attack. So much of the tail load as existed by reason of the dive alone, is calculated in a manner similar to that described in the foregoing discussion on design load for front spar. For so much of the additional load imposed by reason of the inclination of the elevator, it was assumed that the elevator is a flat plate inclined at an angle of 25° to the stabilizer. The viewpoint herein taken, offers the simplest possible method of attack to the problem of pull-up

loads, and so far as the investigation has proceeded, gives good results. It will probably be received as unscientific, but seems to retaliate with the virtues of simplicity and good results. Such a formula permits of the introduction of the two most essential variables for the pull-up condition, viz., the elevator area and the speed at which the maneuver is executed. It only remains then, to establish limiting abrupt pull-up velocities, depending upon the type of aircraft and the type of flying to which the aircraft may be expected to be subjected. This has been done, and these limiting velocities appear under the caption "RESULTS", in this report.

These proposed limiting velocities were plotted against lbs. per H.P. for a large number of airplanes, and it is interesting to note that distinct types grouped themselves together, along a continuous contour, with the exception of the training plane group. It is felt that a reasonable limiting velocity for that type should be about 160 m.p.h. It appears that limiting velocity for abrupt pull-up, is a function of lbs. per H.P.

The distribution of the tail load for the pull-up condition was made in accordance with N.A.C.A. pressure distribution data on the F6C-4 and PW-9 airplanes. Expressions for loads in lbs. per inch of run for both the elevator spar and the stabilizer rear spar have been evolved, and appear in this report on Fig. 2 . This is as far as this analysis could be carried, since the running load on elevator is supported by the elevator hinges, which in turn, impose concentrated loads upon the rear spar of the horizontal stabilizer.

Referring to Fig. 2 , it would be more conservative to consider the load on the leading edge of the stabilizer equal to "w" , instead of "2w" , since such a distribution would impose a more severe load on the rear spar. However, the distribution made in Fig. 2 is in accord with the actual distribution, given in the N.A.C.A. Reports for the tail surfaces of the F6C-4 and PW-9 airplanes.

RESULTS.

DESIGN NORMAL TAIL LOADS FOR FRONT SPAR OF HORIZONTAL STABILIZER:

Pursuit or Fighting Airplanes:

Design normal tail load is that imposed by terminal velocity condition.

$$\text{Normal Tail Load} = K_{M_o} \times A_w \times \frac{C}{d} \times V_t^2$$

where, K_{M_o} = Airfoil K_M at zero lift.

A_w = Wing area.

C = M.A.C.

d = Distance from C.G. To rudder post.

$$V_t = \sqrt{\frac{V_{max}^3 \times W}{375 \times \eta_{max} \times \text{B.H.P.}}}$$

Attack Airplanes:

(As above)

Training Airplanes:

(As above)

Cargo and Bombardment Airplanes:

Design normal tail load is that imposed at a limiting diving speed, which is determined by the character of flying required of the type. The proposed limiting diving speed for these types is 200 m.p.h.

$$\text{Normal Tail Load} = K_{M_o} \times A_w \times \frac{C}{d} \times V^2$$

where, V = Limiting diving speed, which is in turn equal to maximum allowable diving speed, plus a reasonable margin of safety. A 50% margin of safety is suggested, thereby giving an allowable diving speed of 163 m.p.h.

Limiting diving speeds established at this time, should probably be increased from time to time, as power is stepped up.

DESIGN TAIL LOAD FOR ABRUPT PULL-UP CONDITION:

Dynamic Tail Load = Normal tail load at limiting pull-up velocity for dive condition + down load due to inclination of elevator, considered as a flat plate inclined at an angle of 25°.

$$\text{Dynamic Tail Load} = (K_{M_o} \times A_w \times \frac{C}{d} \times V_{lim}^2) + (.00202 \times A_e \times V_{lim}^2)$$

Derivation of formula:

The symbols appearing in so much of the formula as determines the normal tail load due to the dive, are the same as have been described heretofore, for the dive condition. V_{lim} is the design limiting velocity, which in turn is equal to the allowable abrupt pull-up velocity, plus a reasonable margin of safety. The assignment of V_{lim} for any particular type, will depend upon the service expected of the airplane. A_e = elevator area in sq.ft.

$$P_{90^\circ} = .0032 \times A_e \times V_{lim}^2$$

$$\frac{P_{25^\circ}}{P_{90^\circ}} = .7$$

$$P_{25^\circ} = .7 \times .0032 \times A_e \times V_{lim}^2$$

Down load due to inclination of elevator 25° =

$$.7 \times .0032 \times A_e \times V_{lim}^2 \times \cos 25^\circ = .00202 \times A_e \times V_{lim}^2$$

Examples:

F6C-4 Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 190 m.p.h.

$$\text{Dynamic Tail Load} = (.00021 \times \frac{58.7}{167.5} \times 252 \times 190^2) + (.00202 \times 14.8 \times 190^2)$$

$$= (669 + 1080) = 1749 \text{ lbs.}$$

$$\text{Average loading} = \frac{1749}{32.9} = 53.1\#/\text{sq.ft.}$$

DESIGN TAIL LOADS FOR ABRUPT PULL-UP CONDITION:

PW-9 Airplane:

As in the case of the F6C-4 calculation, limiting abrupt pull-up velocity assumed to be 190 m.p.h.

$$\text{Dynamic Tail Load} = (.00020 \times \frac{59.15}{177.3} \times 240.76 \times 190^2) + (.00202 \times 9.9 \times 190^2)$$

$$= (580 + 722) = 1302 \text{ lbs.}$$

$$\text{Average loading} = \frac{1302}{30.3} = 43.0 \ \# \ / \text{ sq.ft.}$$

A-3 Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 160 m.p.h.

$$\text{Dynamic Tail Load} = (.00021 \times \frac{64.3}{200} \times 353 \times 160^2) + (.00202 \times 19.91 \times 160^2)$$

$$= (610 + 1030) = 1640 \text{ lbs.}$$

$$\text{Average loading} = \frac{1640}{47.31} = 34.7 \ \# \ / \text{ sq.ft.}$$

B-2 Airplane:

For this type, limiting pull-up velocity assumed to be 120 m.p.h.

$$\text{Dynamic Tail Load} = (.000246 \times \frac{108.8}{377} \times 120^2) + (.00202 \times 49.6 \times 120^2)$$

$$= (1532 + 1442) = 2974 \text{ lbs.}$$

$$\text{Average loading} = \frac{2974}{146.4} = 20.3 \text{ lbs./ sq.ft.}$$

LB-7 Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 120 m.p.h.

$$\text{Dynamic Tail Load} = (.000228 \times \frac{96.2}{340} \times 1150 \times 120^2) + (.00202 \times 61.2 \times 120^2)$$

$$= (1070 + 1780) = 2850 \text{ lbs.}$$

$$\text{Average loading} = \frac{2850}{144.6} = 19.7 \text{ lbs. / sq. ft.}$$

DESIGN TAIL LOADS FOR ABRUPT PULL-UP CONDITION:

PT-3A Airplane:

For this type, limiting abrupt pull-up velocity assumed to be 160 m.p.h.

$$\text{Dynamic Tail Load} = (.00021 \times \frac{56}{222} \times 300 \times 160^2) + (.00202 \times 16.9 \times 160)$$

$$= (407 + 875) = 1282 \text{ lbs.}$$

$$\text{Average loading} = \frac{1282}{33.6} = 38.1 \text{ lbs. / sq.ft.}$$

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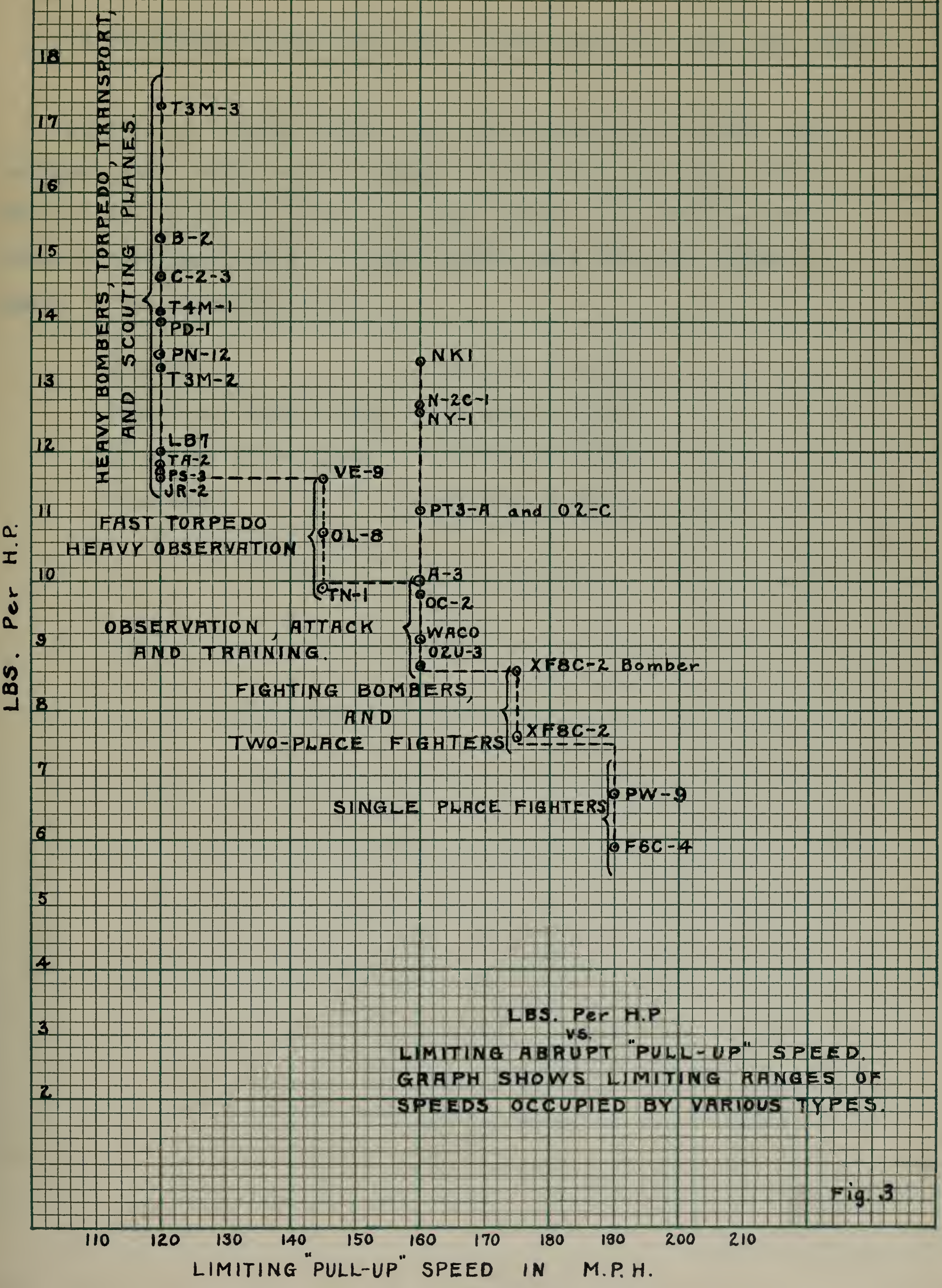

LBS. Per H.P
vs.
LIMITING ABRUPT "PULL-UP" SPEED.
GRAPH SHOWS LIMITING RANGES OF SPEEDS OCCUPIED BY VARIOUS TYPES.

Fig. 3

The following remarks have been copied directly from A.D.M.1061 :

In order to further insure that the construction of the tail surfaces shall never be too flimsy to withstand handling, fabric tension and to resist flutter at high speed, tail surfaces shall never be designed for an average loading less than the values given in the following table:

Pursuit - 50 lbs. / sq.ft.
Attack and Observation - 40 lbs. / sq.ft.
Cargo and Bombardment - 25 lbs. / sq.ft.
Training - 40 lbs. / sq.ft.

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N.A.C.A. Report No. 331 -"Collection of Wind Tunnel Data on Commonly Used Wing Sections"

N.A.C.A. Advance Report -"Tail Surface Loads and Pressures on PW-9 Airplane".

N.A.C.A. reports are prepared by National Advisory Committee for Aeronautics, Washington, D.C.

A.C.I.C. 607 -"Induced Drag of any Biplane".

A.C.I.C. 629 -"Determination of Structural Airplane Drag".

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Handbook of Instructions for Airplane Designers, prepared by U.S.Army Materiel Division, Wright Field, Dayton, Ohio.

APPENDIX:

CURTISS F6C-4 FIGHTING AIRPLANE

TAIL LOAD COMPUTATIONS.

BASIC DATA ON F6C-4 AIRPLANE:

Gross Weight (W) -	2796
Wing Section -	Clark "Y"
M.A.C. (C) -	58.7"
Span, upper -	31.5'
Span, lower -	26.0'
Chord, upper, (average) -	62.6"
Chord, lower, (average) -	49.4"
Gap -	53.31"
Stagger, at L.E., at fuselage -	38.5"
C.G.% M.A.C. -	34.1%
C.G. %M.A.C. below-	35.75%
Area of horizontal tail surfaces -	32.9 sq.ft.
Total wing area -	252 sq.ft.
Distance from C.G. to tail post - (d)	167.5"
Area, upper wing -	158 sq.ft.
Area, lower wing -	94 sq.ft.
Maximum speed, sea level -	157 m.p.h.
B.H.P. 435 at 1950 R.P.M. -	
Diameter of propeller -	8.665 ft.
L.E.M.A.C. 23.6" ahead L.E.L.W.	

Authority: Curtiss Aeroplane & Motor Company, Inc.

CLARK "Y" CHARACTERISTICS:

α	C_L	C_D	C_m 25%	K_y	K_x	K_m 25%
-6.0	-.060	.0108	-.083	-.0001536	.00002765	-.0002125
-4.5	.045	.0107	-.080	.0001152	.0000274	-.000205
-3.0	.167	.0119	-.078	.0004275	.00003045	-.0001997
-1.5	.268	.0139	-.078	.0006860	.0000356	-.0001997
0	.384	.0172	-.070	.0009830	.0000440	-.0001792
1.5	.501	.0228	-.059	.0012820	.00005835	-.000151
3.0	.602	.0288	-.073	.001540	.0000737	-.000187
6.0	.819	.0464	-.079	.002098	.000119	-.0002022
9.0	1.034	.0700	-.054	.00265	.000179	-.0001383
12.0	1.231	.0985	-.036	.00315	.000252	-.0000922
15.0	1.367	.1272	-.077	.00350	.0003255	-.000197
18.0	1.283	.2108	-.056	.003283	.000540	-.0001434
21.0	1.081	.2946	-.048	.00277	.000754	-.000123

Authority; N.A.C.A. Report No. 233. (High Density Tunnel Data).

DETERMINATION OF CORRECTED $\frac{L}{D}$ FOR CELLULE:

Corrected $\frac{L}{D}$ cellule $= \dfrac{1}{\frac{D}{L}\text{ model} + B \times K_y}$

$$B = 125\left(\frac{C_l^2 + C_u^2 + 2s\ Cl\ \ C_u}{A} - \frac{1}{AR\text{ model}}\right)$$

A = (total wing area + area under fuselage) = 264.6 sq.ft.

$$B = 125\left(\frac{4.115^2 + 5.22^2 + 2 \times .53 \times 4.115 \times 5.22}{264.6} - \frac{1}{6}\right)$$
$$= 10.75$$

$\frac{\text{Gap}}{\text{Mean span}} = \frac{4.44}{28.75} = .1545$

$\frac{\text{Lower span}}{\text{Upper span}} = \frac{26.0}{31.5} = .825$

s =.53 (Fig. 4)

Authority: Page 78, Instructions for Airplane Designers,

Note: Area under fuselage assumed to be that of XP3A, given in Air Corps Information Circular No. 629, viz., 12.6 sq.ft.

DETERMINATION OF CORRECTED $\frac{L}{D}$ FOR CELLULE: (Cont'd)

Airfoil α	Airfoil K_y	$\frac{1}{\frac{D}{L}\text{ airfoil} + B \times K_y}$	= Corrected $\frac{L}{D}$ (cellule)
-6.0	-.0001536	$\frac{1}{-.180 + (10.75 \times -.0001536)}$	= -5.5
-4.5	.0001152	$\frac{1}{.238 + (10.75 \times .0001152)}$	= 4.18
-3.0	.0004275	$\frac{1}{.0713 + (10.75 \times .0004275)}$	= 13.18
-1.5	.0006860	$\frac{1}{.0518 + (10.75 \times .000686)}$	= 16.9
0	.0009830	$\frac{1}{.0448 + (10.75 \times .000983)}$	= 18.05
1.5	.0012820	$\frac{1}{.0455 + (10.75 \times .001282)}$	= 16.85
3.0	.001540	$\frac{1}{.0478 + (10.75 \times .00154)}$	= 15.53
6.0	.002098	$\frac{1}{.0567 + (10.75 \times .002098)}$	= 12.6
9.0	.00265	$\frac{1}{.0677 + (10.75 \times .00265)}$	= 10,4
12.0	.00315	$\frac{1}{.080 + (10.75 \times .00315)}$	= 8.78
15.0	.00350	$\frac{1}{.093 + (10.75 \times .00350)}$	= 7.65
18.0	.003283	$\frac{1}{.1644 + (10.75 \times .003283)}$	= 5.01
21.0	.00277	$\frac{1}{.2725 + (10.75 \times .00277)}$	= 3.31

DETERMINATION OF RELATIVE EFFICIENCY OF WINGS - e :

$$s = \frac{\text{chord, lower}}{\text{chord, upper}} = \frac{C_2}{C_1} = \frac{49.4}{62.6} = .79$$

$$\frac{G}{C_1} = \frac{\text{gap}}{\text{chord, upper}} = \frac{53.312}{62.6} = .853$$

$$r = \frac{\text{span, lower}}{\text{span, upper}} = \frac{26.0}{31.5} = .825$$

Stagger = 32.5°

$$e_{(r = .825)} = .825 \; e_{(r = 1)} + \frac{(1-.825)}{(2-Ca_1')} \quad \text{(Fig. 6)}$$

$$e_{(r = 1)} = 1.51 \qquad \text{(Fig. 6)}$$

$$Ca_1' = \frac{\text{Lift of upper wing in biplane}}{\text{Lift of upper wing as monoplane}} = 1.2025 \qquad \text{(Fig. 7)}$$

$$e_{(r = .825)} = (.825 \times 1.51 + \frac{(1 - .825)}{(2 - 1.2025)} = 1.47$$

Authority: A.D.M. 900

DETERMINATION OF EQUIVALENT MONOPLANE ASPECT RATIO OF CELLULE:

A' = area of upper wing = 158 sq.ft.
A^2 = area of lower wings + portion cut out by fuselage,
= 94 + 12.6 = 106.6 sq.ft.

A'' = area of lower wings + 50% of portion cut out by fuselage
= 94 + 6.3 = 100.3 sq.ft.

$$a = \frac{e\,A'}{eA' + A''} = \frac{1.47 \times 158}{(1.47 \times 158) + 100.3} = .70$$

$$\frac{G}{b} = \frac{\text{Gap}}{\text{longer span}} \qquad \frac{4.44}{31.5} = 1.41$$

$$K_{(r=.8)} = 1.054$$

$$K_{(r=.9)} = 1.08$$

$$r = \frac{\text{span, lower}}{\text{span, upper}} = \frac{26.0}{31.5} = .825$$

$$K_{(r = .825)} = 1.06 = \frac{\text{Span of equivalent monoplane}}{\text{Upper span}}$$

Span of equivalent monoplane = (31.5 x 1.06) = 33.4

Equivalent monoplane A.R. $= \frac{(33.4)^2}{(12.6 + 252)} = 4.22$

References: Air Corps I.C. #607 and #629.

DETERMINATION OF CELLULE ANGLES OF ATTACK:

Equivalent monoplane aspect ratio 4.22

$$\alpha^{\circ}_{AR} = \alpha^{\circ}_{AR\ model} - (125 \times 57.3\ K_y)\left(\frac{1}{AR\ model} - \frac{1}{AR}\right)$$

$$\alpha^{\circ}_{4.22} = -6.0 - (125 \times 57.3 \times -.0001536)\left(\frac{1}{6} - \frac{1}{4.22}\right) = -6.08$$

$= -4.5 + (501 \times .0001152) = -4.44$

$= -3.0 + (501 \times .0004275) = -2.78$

$= -1.5 + (501 \times .000686) = -1.15$

$= 0 + (501 \times .000983) = 0.5$

$= 1.5 + (501 \times .001283) = 2.15$

$= 3.0 + (501 \times .00154) = 3.77$

$= 6.0 + (501 \times .0020) = 7.06$

$= 9.0 + (501 \times .00265) = 10.33$

$= 12.0 + (501 \times .00315) = 13.59$

$= 15.0 + (501 \times .0035) = 16.77$

$= 18.0 + (501 \times .00328) = 19.65$

$= 21.0 + (501 \times .00277) = 22.39$

Authority : A.D.M. 1061, page 17.

DETERMINATION OF C.P. :

$$\text{C.P. \% M.A.C.} = .25 - \frac{K_m\ 25\%}{K_n}\ ; \quad \text{where } K_n = K_y \cos\alpha + K_x \sin\alpha$$

Cellule α	K_y	Cellule K_x	K_m about 25% of chord,	K_n	C.P. % M.A.C.
-6.08	-.0001536	.0000279	-.0002125	-.000156	-111.0
-4.44	.0001152	.0000275	-.000205	.000113	206.
-2.78	.0004275	.0000324	-.0001997	.0004245	72
-1.15	.000686	.0000406	-.0001997	.000685	54.1
0.5	.000983	.0000544	-.0001792	.000983	43.2
2.15	.001282	.0000760	-.000151	.001284	36.75
3.77	.001540	.0000992	-.000187	.001548	37.1
7.06	.002098	.0001665	-.0002022	.002106	34.6
10.33	.00265	.000255	-.0001383	.002658	30.2
13.59	.00315	.000359	-.0000922	.00315	27.9
16.77	.00350	.0004575	-.000197	.003484	30.65
19.65	.003283	.000656	-.0001434	.00332	29.3
22.39	.00277	.000837	-.000123	.00288	29.3

By extrapolation of airplane characteristic curves:

-7.0	-.0032	.000029	-.0002125	-.0003215	-41.0
-8.0	-.00049	.000030	-.000213	-.000489	-18.5
-9.0	-.00066	.000031	-.000214	-.000657	- 7.5
-10.0	-.00083	.000037	-.000215	-.000825	- 1.1

REDETERMINATION OF C.P. FROM SMOOTH $K_{M\ L.E.}$:

$$K_r = \sqrt{K_y^2 + K_x^2}$$

Cellule α	Inches from L.E. to C.P.	K_r	$K_{M\ L.E.}$	$K_{M\ L.E.}$ from smooth curve.	C.P. %M.A.C. Redetermined.
-6.08	-65.15	-.000156	-.000173	-.000173	-111.0
-4.44	121.0	.000118	-.000243	-.000243	206
-2.78	42.25	.000429	-.000309	-.000309	72
-1.15	31.75	.000687	-.0003715	-.000 75	54.6
0.5	25.35	.000985	-.000426	-.000445	45.2
2.15	21.55	.001283	-.0004715	-.000505	39.3
3.77	21.8	.001543	-.0005725	-.00057	36.9
7.06	20.3	.002105	-.000728	-.00069	32.8
10.33	17.72	.00266	-.000803	-.000803	30.2
13.59	16.4	.00317	-.000885	-.000916	28.9
16.77	18.0	.00353	-.001082	-.001082	30.65
19.65	17.2	.00335	-.000981	-.000981	29.3
22.39	17.15	.00278	-.000815	-.000815	29.3

By extrapolation of airplane characteristic curves:

-7.0	-24.05	-.000326	-.0001337	-.0001337	-41.0
-8.0	-10.86	-.000491	-.00009075	-.00009075	-18.5
-9.0	- 4.4	-.00066	-.0000495	-.0000495	- 7.5
10.0	.65	-.000831	-.000091	-.0000091	- 1.1

Note: Redetermination of C.P. was made for the purpose of smoothing out basic moment data on Clerk Y airfoil, in order better to define curve of $K_{M\ c.g.}$

DETERMINATION OF $K_{M\ c.g.}$:

Cellule α	$(\beta^\circ - \alpha^\circ)$	Inches from L.C. to C.P.	Inches from C.G. to vector	$K_{M\ c.g.}$
-6.08	4.25°↘	-65.2	83.6	-.000222
-4.44	18.0 °↗	121.0	89.4	-.00018
-2.78	7.25°↗	42.2	19.4	-.000142
-1.15	4.50°↗	32.0	10.2	-.0001193
.5	2.50°↗	26.5	5.5	-.0000923
2.15	1.50°↗	23.05	2.5	-.0000547
3.77	.50°↗	21.65	1.4	-.0000368
7.06	↖2.25°	19.25	.0	0
10.33	↖4.50°	17.72	-0.5	.0000227
13.59	↖7.25°	16.97	-0.25	.0000135
16.77	↖9.50°	18.0	1.5	-.0000902
19.65	↖8.25°	17.2	.2	.0000114
22.39	↖5.50°	17.2	-0.8	.0000379

By extrapolation of airfoil moment curve:

-7.0	↙2.0°	-24.05	44.75	-.0002485
-8.0	↙4.5°	-10.86	32.4	-.000271
-9.0	↙6.25°	- 4.4	26.6	-.000299
10.0	↙7.25°	- 0.65	23.1	-.000327

$(\beta^\circ - \alpha^\circ)$ = Angle between normal to chord and resultant vector, K_r.

DETERMINATION OF AIRPLANE K_y :

Airplane K_y = Cellule K_y + $\frac{\text{Tail Load.}}{A_w \; V^2}$

$\frac{\text{Tail Load}}{A_w \; V^2} = K_M \text{ c.g.} \times \frac{A_w V^2}{A_w \; V^2} \times \frac{c}{d} = K_M \text{ c.g.} \times \frac{58.7}{167.5} = .3504 \; K_M \text{ c.g.}$

Cellule α	(Cellule K_y + .3504 K_M c.g.)	=	Airplane K_y
-6.08	(-.0001536 -.0000781)	=	-.000232
-4.44	(.0001152 -.000064)	=	.0000512
-2.78	(.0004275 -.0000506)	=	.000377
-1.15	(.000686 -.0000401)	=	.000646
0.5	(.000983 -.0000300)	=	.000953
2.15	(.001282 -.0000199)	=	.001262
3.77	(.00154 -.000012)	=	.001528
7.06	(.002098 -.0000038)	=	.002098
10.33	(.00265 0)	=	.00265
13.59	(.00315 0)	=	.00315
16.77	(.00350 -.0000105)	=	.00349
19.65	(.003283 0)	=	.003283
22.39	(.00277 .000015)	=	.00278

By extrapolation of Airplane K_y curve:

-7.0	-.000409
-8.0	-.000587
-9.0	-.000767
-10.0	-.000948

Authority ; A.D.M. 1061

DETERMINATION OF EQUIVALENT FLAT PLATE AREA:

$V_{max} = 157$ m.p.h.

$N = 1950$ r.p.m.

$D = 8.665$ ft.

$$\frac{V}{ND} = \frac{157 \times 1.47}{\frac{1950}{60} \times 8.667} = .82$$

η at max efficiency $= .84$

B.H.P. = 435

$H.P._a = B.H.P. \times \eta = H.P._r$

$H.P._r = 435 \times .84 = 365.$

$$\text{Total Drag} = \frac{375 \times H.P._r}{V} = \frac{375 \times 365}{157} = 872 \text{ lbs.}$$

$W = K_y A V^2$

$2796 = K_y \times 252 \times (157)^2$

Airplane $K_y = .00045 =$ Cellule $K_y + \frac{58.7}{167.5} \times K_{M\ c.g.}$

For Airplane $K_y = .00045$, $\alpha = -2.3$ (from airplane K_y curve).

At $\alpha = -2.3^\circ$, $K_{M\ c.g.} = -.0001375$

Cellule $K_y = .00045 - (.3504 \times -.0001375) = .000498$

Cellule $K_x = .000033$ (from cellule polar).

Wing cellule drag $= .000033 \times 252 \times (157)^2 = 205$ lbs.
Parasite drag $= (872 - 205) = 667$ lbs.

$$A_e \times 32.7 \times \left(\frac{157}{100}\right)^2 = 667$$

$A_e = 8.28$ sq. ft.

DETERMINATION OF AIRPLANE K_x :

Airplane K_x = K_x (cellule) + $\frac{A_e}{A_w}$ x .00327 = K_x (cellule) +

$\frac{8.28}{252}$ x .00327 = K_x (cellule) + .0001075

Cellule α	K_x (cellule)	+	.0001075	=	K_x (airplane)
-6.08	.0000279	+	.0001075	=	.0001354
-4.44	.0000275	+	.0001075	=	.0001350
-2.78	.0000324	+	.0001075	=	.0001399
-1.15	.0000406	+	.0001075	=	.0001481
0.5	.0000544	+	.0001075	=	.0001619
2.15	.000076	+	.0001075	=	.0001835
3.77	.0000992	+	.0001075	=	.0002067
7.06	.0001665	+	.0001075	=	.0002740
10.36	.000255	+	.0001075	=	.0003625
13.59	.000359	+	.0001075	=	.0004665
16.77	.0004575	+	.0001075	=	.000565
19.65	.000656	+	.0001075	=	.0007635
22.39	.000837	+	.0001075	=	.0009445

By extrappolation of Airplane K_x curve:

-7.0	.000136
-8.0	.000137
-9.0	.000138
-10.0	.000144

Authority: A.D.M. 1061. page 16.

DETERMINATION OF VELOCITIES:

$$V = \sqrt{\frac{W}{A}}\ \sqrt{\frac{\cos\beta}{K_y}} = \sqrt{\frac{2796}{252}}\ \sqrt{\frac{\cos\beta}{K_y}} = 3.33\ \sqrt{\frac{\cos\beta}{K_y}}$$

$$= 3.33\ \sqrt{\frac{\sin\beta}{K_x}}$$

Cellule α	Airplane $\frac{L}{D}$	Flight path angle $\beta = \cot^{-1}\frac{L}{D}$	$\cos\beta$	Airplane K_y	$\frac{\cos\beta}{K_y}$	V
-6.08	-1.713	(-)30-16	.8637	-.000232	3725	203.3
-4.7	0	90	sin 1.000	Kx.000135	7400	286.5
-4.44	.379	69-14	.3546	.0000512	6920	277
-2.78	2.69	20-23	.9374	.000377	2485	166
-1.15	4.36	12-55	.9747	.000646	1510	129.5
.5	5.88	9-39	.9858	.000953	1034	107.2
2.15	6.87	8-17	.9896	.001262	784	93.2
3.77	7.34	7-45	.9909	.001528	648	84.1
7.06	7.65	7-27	.9916	.002098	472	72.3
10.33	7.31	7-47	.9908	.00265	374	64.5
13.59	6.75	8-25	.9892	.00315	314	59.4
16.77	6.13	9-16	.9869	.00349	283	56.1
19.65	4.30	13-05	.9740	.003283	297	57.4
22.39	2.945	18-45	.9469	.00278	341	61.5

By extrapolation of Airplane characteristic curves:

-7.0	-3.00	(-)18-26	.9487	-.000409	2320	160.5
-8.0	-4.265	(-)13-12	.9736	-.000587	1660	135.8
-9.0	-5.54	(-)10-14	.9841	-.000767	1283	120
-10.0	-6.56	(-) 8-40	.9886	-.000948	1043	107.5

DETERMINATION OF TAIL LOADS:

$$\text{Normal Tail Load} = K_{M\ c.g.} \times A_W \times \frac{C}{\bar{d}} \times V^2 = K_{M\ c.g.} \times 252 \times \frac{58.7}{167.5} \times V^2$$
$$= 88.3 \times V^2 \times K_{M\ c.g.}$$

$$\text{Dynamic Tail Load} = \text{Normal Tail Load} \times \left(\frac{V_{max}}{V}\right)^2$$

Cellule

α	$K_{M\ c.g.}$	V^2	$\left(\frac{V_{max}}{V}\right)^2$	Dynamic Tail Load.	Normal Tail Load.
-6.08	-.000223	41350	1.987	1620	-815
-4.7	-.0001875	82000	1.0	1359	-1359
-4.44	-.0001825	76700	1.07	1323	-1236
-2.78	-.0001445	27550	2.97	1047	-352
-1.15	-.0001147	16800	4.89	831	-170
.5	-.0000855	11500	7.14	620	-86.8
2.15	-.0000567	8680	9.45	410	-43.4
3.77	-.0000342	7070	11.60	248	-21.4
7.06	-.00001075	5225	15.7	88	- 5.6
10.33	0	3165	19.95	0	0
13.59	0	3530	23.25	0	0
16.77	-.0000301	3140	26.1	218	- 8.3
19.65	0	3300	24.9	0	0
22.39	.0000429	3780	21.7	311	14.3

By extrapolation of Airplane characteristic curves;

-7.0	-.000253	25770	3.185	1775	-575
-8.0	-.000277	18450	4.45	2005	-451
-9.0	-.0003055	14400	5.7	2220	-389
-10.0	-.000336	11560	7.1	2440	-343

Authority: A.D.M. 1061

CURTISS F6C-4 FIGHTING AIRPLANE

TAIL LOAD COMPUTATIONS.

Tail load computations were repeated herein, assuming a gross weight, and C.G. location of the airplane, pressure distribution test of which, are recorded in N.A.C.A. Report No. 307.

Airfoil data forming the basis for these computations have been corrected for Wall Interference.

BASIC DATA ON F6C-4 AIRPLANE:

Gross Weight (W) -	2580
Wing Section -	Clark "Y"
M.A.C. (C) -	58.7"
Span, upper -	31.5'
Span, lower -	26.0'
Chord, upper, (average) -	62.6"
Chord, lower, (average) -	49.4"
Gap -	53.31"
Stagger, at L.E., at fuselage -	38.5"
C.G. % M.A.C. -	33.4%
C.G. %M.A.C. below -	28.1%
Area horizontal tail surfaces -	32.9 sq.ft.
Total Wing Area -	252 sq.ft.
Distance from C.G. to tail post - (d)	167.9"
Area, upper wing -	158 sq.ft.
Area, lower wing -	94 sq.ft.
Maximum speed, sea level -	157 m.p.h.
B.H.P. 435 at 1950 r.p.m.	
Diameter of propeller	8.665 ft.

L.E.M.A.C. 23.6" ahead L.E.L.W.

Note: Gross weight and C.G. location given above, is as specified for F6C-4 Pressure Distribution Tests, recorded in N.A.C.A. Report No. 307.

CLARK "Y" CHARACTERISTICS:

α	C_L	C_D	C_D	$C_{M_{L.E.}}$	C.P.	$\frac{L}{D}$	$\frac{D}{L}$
-6.02	-0.060	.0108	.0106	-.068	-1.117	-5.55	-.180
-4.48	.045	.0107	.0106	-.091	1.99	4.21	.238
-2.94	.167	.0121	.0106	-.120	.720	13.8	.0724
-1.40	.268	.0144	.0106	-.145	.541	18.6	.0537
.15	.384	.0182	.0103	-.166	.432	21.1	.0474
1.69	.501	.0245	.0111	-.185	.368	20.4	.0489
3.23	.602	.0312	.0119	-.224	.371	19.3	.0518
6.31	.819	.0508	.0152	-.284	.346	16.1	.062
9.39	1.034	.0770	.0201	-.312	.302	13.4	.0745
12.47	1.231	.1085	.0280	-.360	.294	11.4	.0881
15.52	1.367	.1395	.0403	-.415	.306	9.8	.102
18.49	1.283	.2217	.1342	-.378	.294	5.8	.1725
21.41	1.081	.3025	.2402	-.328	.293	3.58	.2795

Authority: N.A.C.A. Report No. 331 (High Density Tunnel, data corrected for Wall Interference).

DETERMINATION OF CORRECTED $\frac{L}{D}$ FOR CELLULE :

Airfoil

α	K_y	$\frac{1}{\frac{D}{L}\text{airfoil} + B \times K_y}$	= corrected $\frac{L}{D}$ (cellule)
-6.02	-.0001536	$\frac{1}{-.180+(10.75 \times -.0001536)}$	= -5.5
-4.48	.0001152	$\frac{1}{.238+(10.75 \times .0001152)}$	= 4.18
-2.94	.0004275	$\frac{1}{.0724+(10.75 \times .0004275)}$	= 13.0
-1.40	.000686	$\frac{1}{.0537+(10.75 \times .000686)}$	= 16.38
.15	.000983	$\frac{1}{.0474+(10.75 \times .000983)}$	= 17.24
1.69	.001282	$\frac{1}{.498+(10.75 \times .001282)}$	= 15.95
3.23	.001540	$\frac{1}{.518+(10.75 \times .001540)}$	= 14.63
6.31	.002098	$\frac{1}{.062+(10.75 \times .002098)}$	= 11.84
9.39	.00265	$\frac{1}{.0745+(10.75 \times .002650)}$	= 9.7
12.47	.003150	$\frac{1}{.0881+(10.75 \times .003150)}$	= 8.2
15.52	.00350	$\frac{1}{.102+(10.75 \times .00350)}$	= 7.17
18.49	.003283	$\frac{1}{.1725+(10.75 \times .003283)}$	= 4.81
21.41	.002770	$\frac{1}{.2795+(10.75 \times .00277)}$	= 3.23

Note: Value of the factor "B" same as for previous calculation for F6C-4 Airplane.

DETERMINATION OF CELLULE ANGLES OF ATTACK:

Equivalent monoplane aspect ratio 4.22

$$\alpha^{\circ}_{AR} = \alpha^{\circ}_{AR\ model} - (125 \times 57.3\ K_y)\left(\frac{1}{AR\ model} - \frac{1}{AR}\right)$$

$$\alpha^{\circ}_{4.22} = -6.02 - (125 \times 57.3 \times -.0001536)\left(\frac{1}{6} - \frac{1}{4.22}\right) = -6.1$$

$$-4.48 + (501 \times .0001152) = -4.42$$

$$-2.94 + (501 \times .0004275) = -2.72$$

$$-1.4 + (501 \times .000686) = -1.05$$

$$.15 + (501 \times .000983) = .65$$

$$1.69 + (501 \times .001282) = 2.34$$

$$3.23 + (501 \times .001540) = 4.00$$

$$6.31 + (501 \times .002098) = 7.37$$

$$9.39 + (501 \times .002650) = 10.72$$

$$12.47 + (501 \times .003150) = 14.06$$

$$15.52 + (501 \times .00350) = 17.29$$

$$18.49 + (501 \times .003282) = 20.14$$

$$21.41 + (501 \times .00277) = 22.8$$

Authority: A.D.M. 1061, page 17.

DETERMINATION OF $K_{m_{c.g.}}$

Cellule α	K_y	Cellule K_x	K_r	$(\beta^\circ - \alpha^\circ)$	Inches to C.P. from M.A.C.	Inches from L.E.C.G. to vector	$K_{m_{c.g.}}$
-6.1	-.0001536	.0000279	-.000155	4¾ ↘	65.5	83.5	-.0002205
-4.42	.0001152	.0000275	.000118	17¾ ↗	116.7	87.4	-.0001758
-2.72	.0004275	.0000329	.000429	7 ↗	42.25	19.4	-.000142
-1.05	.000686	.0000418	.000687	4¾ ↗	31.75	10.7	-.0001252
.65	.000983	.0000570	.000985	2¾ ↗	25.4	4.9	-.0000822
2.34	.001282	.0000803	.001284	1¼ ↗	21.6	1.5	-.0000328
4.00	.001540	.0001052	.001544	0	21.8	2.1	-.00005525
7.37	.002098	.000177	.002108	↖ 2½	20.3	1.4	-.0000502
10.72	.00265	.000273	.002656	↖ 4¾	17.73	-0.4	.0000181
14.06	.00315	.000384	.003172	↖ 6¾	17.26	-0.4	.0000216
17.29	.00350	.000488	.003536	↖ 9¼	17.97	0.9	-.0000543
20.14	.003283	.000683	.003352	↖ 8	17.26	.0	0

$(\beta^\circ - \alpha^\circ)$ = Angle between normal to chord and resultant vector,
$K_r = \sqrt{K_y^2 + K_x^2}$

By extrapolation of $K_{m_{c.g.}}$ curve:

-7.0	-.0002475
-8.0	-.000277
-9.0	-.000305
-10.0	-.00336

DETERMINATION OF AIRPLANE K_y :

$$\text{Airplane } K_y = \text{Cellule } K_y + \frac{\text{Tail Load}}{A_w V^2}$$

$$\frac{\text{Tail Load}}{A_w V^2} = K_{mc.g.} \times \frac{A_w V^2}{A_w V^2} \times \frac{C}{d} = K_{mc.g.} \times \frac{58.7}{167.9} = .3495\ K_{m_{c.g.}}$$

Cellule α	(Cellule K_y + .3495 $K_{m_{c.g.}}$)	= Airplane K_y
-6.10	(-.0001536 - .3495 x .0002205) =	-.0002307
-4.42	(.0001152 - .3495 x .0001758) =	.0000537
-2.72	(.0004275 - .3495 x .000142) =	.000378
-1.05	(.000686 - .3495 x .000110) =	.000648
.65	(.000983 - .3495 x .0000822) =	.0009543
2.34	(.001282 - .3495 x .000055) =	.001263
4.00	(.001540 - .3495 x .000040) =	.001526
7.37	(.002098 - .3495 x .000015) =	.002093
10.72	(.00265 - .3495 x .000002) =	.00265
14.06	(.00315 - .3495 x .000002) =	.00315
17.29	(.00350 - .3495 x .000020) =	.00350
20.14	(.003283 - .3495 x .00005) =	.003266

Note: Values of $K_{m_{c.g.}}$ above, taken from faired curve.

By extrapolation of Airplane K_y curve.

-7.0	-.00041
-8.0	-.00059
-9.0	-.00077
-10.0	-.00095

DETERMINATION OF AIRPLANE K_x :

Airplane $K_x = K_x$(cellule) $+ \frac{Ae}{Aw} \times .00327 = K_x$(cellule) $+ \frac{8.28}{252} \times .00327$

$= Kx$(cellule) $+ .0001075$

Cellule α	K_x(cellule)	+ .0001075	= K_x (airplane.
-6.1	.0000279	+ .0001075	= .0001354
-4.42	.0000275	+ .0001075	= .0001350
-2.72	.0000329	+ .0001075	= .0001404
-1.05	.0000418	+ .0001075	= .0001493
.65	.0000570	+ .0001075	= .0001645
2.34	.0000803	+ .0001075	= .0001878
4.00	.0001052	+ .0001075	= .0002127
7.37	.000177	+ .0001075	= .0002845
10.72	.000273	+ .0001075	= .0003805
14.06	.000384	+ .0001075	= .0004915
17.29	.000488	+ .0001075	= .0005955
20.14	.000683	+ .0001075	= .0007905

By extrapolation of Airplane K_x curve.

-7.0	.000136
-8.0	.000137
-9.0	.000138
-10.0	.000142

DETERMINATION OF VELOCITIES:

$$V = \sqrt{\frac{W}{A}}\ \sqrt{\frac{\cos\beta}{K_y}} = \sqrt{\frac{2580}{252}}\ \sqrt{\frac{\cos\beta}{K_y}} = 3.2\ \sqrt{\frac{\cos\beta}{K_y}}$$

β = Flight path angle.

$$= 3.2\ \sqrt{\frac{\sin\beta}{K_x}}$$

Cellule α	Airplane $\frac{L}{D}$	Flight Path angle $\beta = \cot^{-1}\frac{L}{D}$	$\cos\beta$	Airplane K_y	$\frac{\cos\beta}{K_y}$	V
-6.1	-1.70	(-) 30°-28′	-.8619	-.0002307	3740	196
-4.7	0	90°	Sin 1.000	K_x .000135	7400	275
-4.42	.398	68°-18′	.3697	.0000537	6880	266
-2.72	2.69	20°-24′	.9373	.000378	2480	159.5
-1.05	4.34	12°-59′	.9744	.000648	1504	124
.65	5.8	9°-47′	.9855	.0009543	1034	101.5
2.34	6.72	8°-28′	.9891	.001263	783	89.6
4.00	7.17	7°-56′	.9904	.001526	649	81.5
7.37	7.36	7°-44′	.9909	.002093	473	69.6
10.72	6.96	8°-11′	.9898	.00265	373	61.8
14.06	6.40	8°-53′	.9880	.00315	314	56.7
17.29	5.88	9°-39′	.9859	.00350	282	53.7
20.14	4.13	13°-37′	.9719	.003266	297	55.2

By extrapolation of Airplane characteristic curves:

-7	-3.01	(-) 18°-23′	-.9490	-.00041	2310	154
-8	-4.3	(-) 13°-05′	-.9740	-.00059	1650	130
-9	-5.58	(-) 10°-09′	-.9843	-.00077	1280	114.5
-10	-6.6	(-) 8°-37′	-.9887	-.00095	1040	103

EFFECT OF C.G. LOCATION ON VALUE $K_{M\ c.g.}$ AT ZERO LIFT:

(4)

C.G. at (25% M.A.C.
(On M.A.C.

Cellule α	Inches from C.G. to vector	K_r	K_{M} c.g.
-7.0	38.7	-.000326	-.000215
-6.08	79.6	-.000156	-.000212
-4.44	101.0	.000118	-.000203
-2.78	27.2	.000429	-.000199
-1.15	17.3	.000687	-.0002025

$K_{M\ c.g.}$ at zero lift = -.000205

EFFECT ON VALUE OF NORMAL TAIL LOAD OF USING AIRFOIL K_M AT ZERO LIFT FOR $K_{M\ c.g.}$ AT VARIOUS C.G. LOCATIONS.

CLARK "Y" , K_M at zero lift = $-.00021$

(5) C.G. at 34.1% M.A.C.

Normal Tail Load $= \frac{58.7}{167.5} \times 252 \times 82000 \times -.00021 = 1520$

(6) C.G. at 25% M.A.C.

Normal Tail Load $= 58.7 \times 252 \times 82000 \times -.00021 = 1475$

(7) C.G. at 40% M.A.C.

Normal Tail Load $= \frac{58.7}{164.} \times 252 \times 82000 \times -.00021 = 1550$

DETERMINATION OF TAIL LOADS:

$$\text{Normal Tail Load} = K_{M_{c.g.}} \times A_w \times \frac{C}{d} \times V^2 = K_{M_{c.g.}} \times 252 \times \frac{58.7}{167.9} \times V^2$$

$$= 88.1 \times V^2 \times K_{M_{c.g.}}$$

$$\text{Dynamic Tail Load} = \text{Normal Tail Load} \times \left(\frac{V_{max}}{V}\right)^2$$

α	$K_{M_{c.g.}}$	V^2	Normal Tail Load	$\left(\frac{V_{max}}{V}\right)^2$	Dynamic Tail Load.
-6.1	-.0002205	38416	-746	1.97	1470
-4.7	-.000185	75600	-1232	1.00	1232
-4.42	-.0001758	70750	-1095	1.07	1172
-2.72	-.000142	25450	-318	2.97	945
-1.05	-.000110	15376	-148.7	4.92	731
0.65	-.0000822	10300	- 74.5	7.56	563
2.34	-.000055	8028	- 38.8	9.43	366
4.00	-.000040	6642	- 23.4	11.4	267
7.37	-.000015	4844	- 6.4	15.6	99.8
10.72	-.000002	3819	- 0.7	19.8	13.3
14.06	-.000002	3215	- 0.6	23.5	13.3
17.29	-.000020	2884	- 5.1	26.2	133
20.14	-.000050	3047	- 13.4	24.8	332

By extrapolation of Airplane characteristic curves:

-7.0	-.0002475	23650	-516	3.18	1640
-8.0	-.000277	16900	-412	4.47	1840
-9.0	-.000305	13220	-355	5.76	2045
-10.0	-.000336	10609	-314	7.12	2235

EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD.

AT ZERO LIFT:

$$\text{Normal Tail Load} = K_{M\ c.g.} \times A_w \times \frac{C}{d} \times V^2$$

(1) C.G. at (40% M.A.C.
(25% below M.A.C.

$d = 167.5 - (.40 - .341) \times 58.7 = (167.5 - 3.46) = 164''$

$$\text{Normal Tail Load} = K_{M\ c.g.} \times 252 \times \frac{58.7}{164.} \times V^2 = 90.15 \times V^2 \times K_{M\ c.g.}$$

Cellule α	$(90.15 \times V^2 \times K_{M\ c.g.})$	= Normal Tail Load.
-7.0	$(90.15 \times 25770 \times -.0002695)$	= -625.
-6.08	$(90.15 \times 41350 \times -.00023)$	= -857.2
-4.7	$(90.15 \times 82000 \times -.000178)$	= -1315
-4.44	$(90.15 \times 76700 \times -.00017)$	= -1176
-2.78	$(90.15 \times 27550 \times -.000111)$	= - 276
-1.15	$(90.15 \times 16800 \times -.0000737)$	= - 111.7

(2) C.G. at (40% M.A.C.
(On M.A.C.

$d = 164''$

$$\text{Normal Tail Load} = 90.15 \times V^2 \times K_{M\ c.g.}$$

Cellule α		
-7.0	$(90.15 \times 25770 \times -.000264)$ =	- 612.5
-6.08	$(90.15 \times 41350 \times -.000235)$ =	- 876
-4.75	$(90.15 \times 82000 \times -.000193)$ =	-1426
-4.44	$(90.15 \times 76700 \times -.000186)$ =	-1286
-2.78	$(90.15 \times 27550 \times -.0001353)$ =	- 336
-1.15	$(90.15 \times 16800 \times -.0000984)$ =	- 149

COMPARISON OF AIRFOIL MOMENT COEFFICIENT AT ZERO LIFT AND K_M c.g. AT ZERO LIFT FOR SEVERAL AIRPLANES:

F6C-4 Airplane:

Clark "Y" airfoil.

$K_{Mo} = -.00021$ (N.A.C.A.) (M.I.T.)

K_M c.g. $= -.0001875$ (Fig. 12)

PW-9 Airplane:

Gott. 436 airfoil.

$K_{Mo} = -.00020$ (M.I.T.)

K_M c.g. $= -.000189$ (A.D.M. 1061)

B-2 Airplane:

C-72 airfoil.

$K_{Mo} = -.000246$ (A.D.M. 1061)

K_M c.g. $= -.000235$ (A.D.M. 1061)

A-3 Airplane:

Clark "Y" airfoil.

$K_{Mo} = -.00021$ (N.A.C.A.)

K_M c.g. $= -.000172$ (A.D.M. 1061)

PT-3A Airplane:

Clark "Y" Airfoil.

$K_{Mo} = -.00021$ (N.A.C.A.)

K_M c.g. $= -.00021$ (A.D.M. 1061)

EFFECT OF C.G. LOCATION ON VALUE OF $K_{M_{c.g.}}$ AT ZERO LIFT:

(1)

C.G. at (40% M.A.C.
(45% below M.A.C.

Cellule α	Inches from C.G. to vector	K_r	K_M c.g.
-7.0	48.5	-.000326	-.0002695
-6.08	86.5	-.000156	-.00023
-4.44	84.5	.000118	-.00017
-2.78	15.2	.000429	-.000111
-1.15	6.5	.000687	-.0000737

K_M c.g. at zero lift = -.000178

(2)

C.G. at (40% M.A.C.
(on M.A.C.

Cellule	Inches from C.G. to vector	K_r	K_M c.g.
-7.0	47.5	-.000326	-.000264
-6.08	88.4	-.000156	-.000235
-4.44	92.6	.000118	-.000186
-2.78	18.5	.000429	-.0001353
-1.15	8.4	.000687	-.0000984

K_m c.g. at zero lift = -.000193

(3)

C.G. at (25% M.A.C.
(45% below M.A.C.

Cellule	Inches from C.G. to vector	K_r	K_M c.g.
-7.0	39.7	-.000326	-.0002205
-6.08	77.7	-.000156	-.0002065
-4.44	92.7	.000118	-.0001864
-2.78	23.9	.000429	-.000175
-1.15	15.1	.000687	-.000177

K_M c.g. at zero lift = -.000189

EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD AT ZERO LIFT:

(3) C.G. at (25% M.A.C.
(45% below M.A.C.

$d = 167.5 + (.341-.25)x58.7 = 167.5 + 5.29 = 172.8$

Normal Tail Load $= K_{M\ c.g.} x\ 252\ x\ \frac{58.7}{172.8}\ x V^2 = 85.5 x V^2 x K_{M\ c.g.}$

Cellule α	$(85.5\ x\ V^2 x\ K_{M\ c.g.})$	=	Normal Tail Load.
-7.0	(85.5 x 25770 x -.0002205)	=	- 486
-6.08	(85.5 x 41350 x -.0002065)	=	- 730
-4.7	(85.5 x 82000 x -.000189)	=	-1326
-4.44	(85.5 x 76700 x -.0001865)	=	-1224
-2.78	(85.5 x 27550 x -.000175)	=	- 412
-1.15	(85.5 x 16800 x -.000177)	=	- 254

(4) C.G. at (25% M.A.C.
(On M.A.C.

d = 172.8"

-7.0	(85.5 x 25770 x -.000215)	=	- 474
-6.08	(85.5 x 41350 x -.000212)	=	- 750
-4.7	(85.5 x 82000 x -.000205)	=	-1438
-4.44	(85.5 x 76700 x -.000203)	=	-1330
-2.78	(85.5 x 27550 x -.000199)	=	- 469
-1.15	(85.5 x 16800 x -.0002025)	=	- 291

COMPARISON OF AIRFOIL MOMENT COEFFICIENT AT ZERO LIFT AND K_M c.g. AT ZERO LIFT, FOR SEVERAL AIRPLANES:

C-2 Airplane:

Fokker airfoil.

K_{Mo} = Not recorded.

K_M c.g. = -.0001614 (A.D.M. 1061)

LB-7 Airplane.

Gott. 398 airfoil.

K_{Mo} = -.000228 (W.N.Y.)

K_M c.g. = -.00022 (A.D.M. 1061)

COMPARATIVE TERMINAL VELOCITY CALCULATIONS:

Data for the calculations of terminal velocity from the drag formula, have been taken from A.D.M. 1061, except for the F6C-4 airplane. Data for the calculations of terminal velocity from the power absorbed formula, have been taken from A.C.I.C. #629. except for the case of the F6C-4 airplane.

F6C-4 Airplane:

$W \cos\beta = K_x A_w V^2$ β=flight path angle = 90°

(1) $V = \sqrt{\dfrac{W}{K_x A_w}} \sqrt{\dfrac{2796}{.000135 \times 252}} = 286.5$ m.p.h.

(2) $V = \sqrt{\dfrac{V_{max}^3 \times W}{375 \times \eta_{max} \times B.H.P.}} = \sqrt{\dfrac{(157)^3 \times 2796}{375 \times .84 \times 435}} = 281$ m.p.h.

PW-9 Airplane:

(1) $V = \sqrt{\dfrac{2890}{.0001227 \times 240.76}} = 312$ A.D.M. 1061

(2) $V = \sqrt{\dfrac{(165.5)^3 \times 2890}{375 \times .831 \times 431}} = 312$

B-2 Airplane:

(1) $V = \sqrt{\dfrac{16610}{.000118 \times 1498}} = 306$

(2) $V = \sqrt{\dfrac{(129.85)^3 \times 16610}{375 \times .816 \times 1227}} = 311$

C-2 Airplane:

(1) $V = \sqrt{\dfrac{10393}{.00017 \times 747}} = 286$

(2) $V = \sqrt{\dfrac{(116)^3 \times 10393}{375 \times .799 \times 671}} = 284$

COMPARATIVE TERMINAL VELOCITY CALCULATIONS: (cont'd)

A-3 Airplane:

(1) $V = \sqrt{\dfrac{4377}{.0001338 \times 353}} = 304.5$

(2) $V = \sqrt{\dfrac{(141.4)^3 \times 4377}{375 \times .806 \times 440}} = 305$

PT-3A Airplane:

(1) $V = \sqrt{\dfrac{2431}{.000189 \times 300}} = 207$

(2) $V = \sqrt{\dfrac{(105.5)^3 \times 2431}{375 \times .796 \times 220}} = 208$

LB-7 Airplane:

(1) $V = \sqrt{\dfrac{12868}{.0001605 \times 1150}} = 264$

(2) $V = \sqrt{\dfrac{(117.6)^3 \times 12868}{375 \times .784 \times 1070}} = 258$

COMPARATIVE MAXIMUM NORMAL TAIL LOAD CALCULATIONS:

(1) Formula: Max. Normal Tail Load $= K_{M\,c.g.} \times \frac{C}{d} \times A_W \times V_t^2$

V_t from, $W \cos\beta = K_x \; A_W \; V_t^2$

(2) Formula: Max. Normal Tail Load $= K_{M_o} \times \frac{C}{d} \times A_W \times V_t^2$

where K_{M_o} = Airfoil K_M at zero lift,

and , $V_t = \sqrt{\frac{V^3_{max} \times W}{375 \times \eta_{max} \times B.H.P.}}$

F6C-4 Airplane:

(1) Max. Normal Tail Load = -1359 lbs. (page 47)

(2) Max. Normal Tail Load $= -.00021 \times \frac{58.7}{167.5} \times 252 \times (281)^2 = -1465$

PW-9 Airplane:

(1) Max. Normal Tail Load = -1480 (A.D.M. 1061)

(2) Max. Normal Tail Load $= -.00020 \times \frac{59.15}{177.3} \times 240.76 \times (312)^2$

$= -1565$

B-2 Airplane:

(1) Max. Normal Tail Load = - 9500 (A.D.M. 1061)

(2) Max. Normal Tail Load $= -.000246 \times \frac{108.8 \times 1498}{377} \times (311)^2$

$= -10300$

A-3 Airplane:

(1) Max. Normal Tail Load = -1920 (A.D.M. 1061)

(2) Max. Normal Tail Load $= -.00021 \times \frac{64.3}{200} \times 353 \times (305)^2 = -2220$

COMPARATIVE MAXIMUM TAIL LOAD CALCULATIONS: (cont'd)

PT-3A Airplane:

(1) Max. Normal Tail Load = -680 (A.D.M. 1061)

(2) Max. Normal Tail Load $= -.00021 \times \frac{56}{222} \times 300 \times (208)^2$

$= -687$

C-2 Airplane:

(1) Max. Normal Tail Load = -3160 (A.D.M. 1061)

(2) Max. Normal Tail Load $= \frac{128.9}{402} \times 747 \times (284)^2 \times K_{M_0}^{*} =$

LB-7 Airplane:

(1) Max: Normal Tail Load = -5000 (A.D.M. 1061)

(2) Max. Normal Tail Load $= -.000228 \times \frac{96.2}{340} \times 1150 \times (258)^2$

$= -4940$

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* Not recorded.

COMPARATIVE VALUES OF MAXIMUM NORMAL TAIL LOAD:

Airplane	K_M c.g. at zero lift.	Airfoil K_M at zero lift.	V_t (A.D.M. 1061)	V_t by formula (2) below	Max. normal Tail Load by (A.D.M. 1061)	Max. Normal Tail Load by proposed modification
F6C-4	-.0001875	-.00021	286.5	281	1359	1465
PW-9	-.000189	-.00020	312	312	1480	1565
B-2	-.000235	-.000246	306	311	9500	10300
A-3	-.000182	-.00021	304.5	305	1920	2220
PT-3A	-.00021	-.00021	207	208	680	687
C-2	-.0001614	———	286	284	3160	———
LB-7	-.00022	-.000228	264	258	5000	4940

(1) $W \cos 90^\circ = K_x \, A_w \quad V_t^2$ (A.D.M. 1061)

(2) $V = \sqrt{\dfrac{V_{max}^3 \times W}{375 \times \eta_{max} \times B.H.P.}}$ (power absorbed formula)

BOEING PW-9 FIGHTING AIRPLANE

TAIL LOAD COMPUTATIONS.

BASIC DATA ON PW-9 AIRPLANE:

Gross Weight	2890 lbs.
Wing Section--	Cott., 436
M.A.C. (C)--	59.15"
Span , upper--	32.00'
Span , lower--	22.50'
Chord, upper, (average)--	5.13'
Chord, lower, (average)--	4.51'
Gap---	4.30'
Stagger, at leading edge of fuselage---	5.1°
C.G. % M.A.C.---	31.
C.G. % M.A.C. below---	35.
Area of horizontal tail surfaces--	30.3 sq.ft.
Total wing area---	240.75 sq.ft.
Distance from C.G. to tail post -(d)--	177.3"
Area of upper wing---	159.8 sq.ft.
Area of lower wing---	80.96 sq.ft.
Maximum speed at sea level--	165.5 M.P.H.
B.H.P.	431.0
Diameter of propeller---	8.67 ft.

Authority: Air Corps Information Circular No. 629

GOTT. 436 AIRFOIL CHARACTERISTICS:

α°	C_L	C_D	C_m	K_y	K_x	K_m
-8.9	-.236	.0437	-0.009	-.000604	.000112	-.000023
-6.0	-.050	.0144	-0.063	-.000128	.00003685	-.000161
-4.5	.050	.0130	-0.084	.000128	.0000333	-.000215
-3.0	.150	.0133	-0.107	.000384	.00003405	-.000274
-1.6	.246	.0159	-0.130	.000630	.0000407	-.000333
-0.1	.349	.0189	-0.154	.000894	.0000484	-.000394
1.3	.451	.0247	-0.182	.001153	.0000632	-.000466.
2.8	.548	.0294	-0.202	.001405	.0000753	-.000517
4.3	.647	.0382	-0.226	.001657	.0000978	-.000579
5.7	.751	.0488	-0.248	.001922	.000125	-.000635
8.7	.945	.0728	-0.301	.00242	.0001866	-.00077
11.6	1.120	.0999	-0.343	.00287	.000254	-.000878
14.6	1.204	.138	-0.365	.00308	.0003535	-.000935

Authority: N.A.C.A. Report No. 233. (Gottingen Tunnel Data).

DETERMINATION OF CORRECTED $\frac{L}{D}$ FOR CELLULE:

$$\text{Corrected } \frac{L}{D} \text{ cellule} = \frac{1}{\frac{D}{L}\text{model} + B \times K_y}$$

$$B = \left(\frac{C_l^2 + C_u^2 + 2s \quad C_l \quad C_u}{A} - \frac{1}{\text{AR model}} \right)$$

$$A = (\text{Total wing area} + \text{area under fuselage}) = 252.76 \text{ sq.ft.}$$

$$B = 125 \left(\frac{\overline{4.51}^2 + \overline{5.13}^2 + 2 \times .48 \times 4.51 \times 5.13}{252.76} - \frac{1}{5} \right)$$

$$= 9.00$$

$$\frac{\text{Gap}}{\text{Mean span}} = \frac{4.30}{27.25} = 15.75$$

$$\frac{\text{Lower span}}{\text{upper span}} \quad \frac{22.5}{32.0} = .703$$

$$s = .48$$

Authority: Page 78, Instructions for Airplane Designers.

DETERMINATION OF CORRECTED $\frac{L}{D}$ FOR CELLULE: (Cont'd)

Airfoil α°	Airfoil K_y	$\frac{1}{\frac{D}{L}\text{ model} + B \times K_y}$	$= \frac{L}{D}$ Corrected (cellule)
-8.9	-.000604		-5.25
-6.0	-.000128		-3.485
-4.5	.000128		3.85
-3.0	.000384		10.88
-1.6	.000630		14.25
-0.1	.000894		16.12
1.3	.001153		15.40
2.8	.001405		15.05
4.3	.001657		13.50
5.7	.001922		12.16
8.7	.00242		10.13
11.6	.00287		8.68
14.6	.00308		7.025

DETERMINATION OF CELLULE ANGLES OF ATTACK:

Equivalent monoplane aspect ratio 4.16

$$\alpha^\circ_{AR} = \alpha^\circ_{AR\text{ model}} - (125 \times 57.3\, K_y)\left(\frac{1}{AR\text{ model}} - \frac{1}{AR}\right)$$

$$\alpha^\circ_{4.16} = -8.9 - (125 \times 57.3 \times -.000604)\left(\frac{1}{5} - \frac{1}{4.16}\right) = -9.07$$

-6.0	-6.04
-4.5	-4.46
-3.0	-2.89
-1.6	-1.415
-0.1	.158
1.3	1.63
2.8	3.205
4.3	4.775
5.7	6.25
8.7	9.395
11.6	12.43
14.6	15.49

DETERMINATION OF K_M c.g. :

Cellule α	$(\beta^\circ - \alpha^\circ)$	Inches from L.E. to C.P.	Inches from C.G. to vector.	K_M c.g.
-9.07	↘ 1.5	2.21	20.0	-.000208
-6.04	↘10.0	72.15	85.8	-.000196
-4.46	19.0 ↗	101.8	70.7	-.000158
-2.89	8.0 ↗	42.5	21.0	-.000137
-1.42	5.25 ↗	31.4	10.8	-.0001152
.16	3.5 ↗	26.0	6.25	-.0000946
1.63	2.0 ↗	23.07	4.0	-.0000785
3.21	.5 ↖	21.3	2.8	-.0000665
4.78	.75 ↖	20.12	2.0	-.0000563
6.26	1.5 ↖	18.92	1.2	-.0000390
9.4	4.0 ↖	18.35	1.5	-.0000616
12.43	5.25 ↖	17.75	1.5	-.0000735
15.49	7.25 ↖	18.35	2.6	-.0001368

$(\beta^\circ - \alpha^\circ)$ = Angle between normal to chord and resultant vector, K_r

$$K_M \text{ c.g.} = \frac{(\text{Distance from C.G. to vector}) \times K_r}{\text{Chord}}$$

DETERMINATION OF AIRPLANE K_y :

$$\text{Airplane } K_y = \text{Cellule } K_y + \frac{\text{Tail Load}}{A_w \ V^2}$$

$$\frac{\text{Tail Load}}{A_w \ V^2} = K_M \text{ c.g.} \times \frac{A_w V^2}{A_w V^2} \times \frac{c}{d} = K_M \text{ c.g.} \times \frac{59.15}{177.5} = .333 \ K_M \text{ c.g.}$$

Cellule α	(Cellule K_y + .333 K_M c.g.) =	Airplane K_y
-9.07		-.000673
-6.04		-.0001933
-4.46		.000075
-2.89		.000338
-1.42		.000595
.16		.0008625
1.63		.0011268
3.21		.001383
4.78		.00164
6.26		.001909
9.4		.002404
12.43		.00284
15.49		.003034

Authority: A.D.M. 1061

DETERMINATION OF AIRPLANE K_x :

$$\text{Airplane } K_x = K_x \text{ (cellule)} + \frac{A_e}{A_w} \times .00327 = K_x \text{ (cellule)} + \frac{5.64}{240.76} \times .00327 = K_x \text{ (cellule)} + .0000765$$

Cellule α	K_x (cellule) + .0000765 =	K_x (airplane)
-9.07		.0001915
-6.04		.0001132
-4.46		.0001099
-2.89		.0001118
-1.42		.0001207
.16		.0001319
1.63		.0001515
3.21		.0001698
4.78		.0001991
6.26		.0002345
9.4		.0003155
12.43		.0004075
15.49		.0005155

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DETERMINATION OF VELOCITIES:

$$V = \sqrt{\frac{W}{A}} \sqrt{\frac{\cos\beta}{K_y}} = \sqrt{\frac{2890}{240.76}} \sqrt{\frac{\cos\beta}{K_y}} = 3.46 \sqrt{\frac{\cos\beta}{K_y}}$$

Cellule α	Airplane $\frac{L}{D}$	Flight path angle. $\beta = \cot^{-1}\frac{L}{D}$	$\cos\beta$	Airplane K_y	$\frac{\cos\beta}{K_y}$	V
-9.07	3.51	15.9	.9617	-.000673	1430	131
-6.04	1.705	30.4	.8625			231.5
-4.46	.683	55.67	.5640			300
-2.89	3.02	18.3	.9494			183
-1.42	4.935	11.45	.9801			140
.16	6.55	8.67	.9886			117.2
1.63	7.43	7.625	.9912			102.8
3.21	8.14	7.0	.9925			92
4.78	8.24	6.92	.9928			85
6.26	8.13	7.01	.9925			79
9.4	7.62	7.48	.9915			70.3
12.43	6.97	8.17	.9922			64.7
15.49	5.87	9.67	.9858			62.4

DETERMINATION OF TAIL LOADS:

Normal Tail Load $= K_{M_{cg}} \times A_W \times \frac{C}{d} \times V^2 = K_{M_{cg}} \times V^2 \times 80.25$

Dynamic Tail Load $=$ Normal Tail Load $\times \left(\frac{V_{max}}{V}\right)^2$

Cellule $\alpha°$	$K_{M_{cg}}$	V^2	$\left(\frac{V_{max}}{V}\right)^2$	Dynamic Tail Load	Normal Tail Load.
-9.07	-.000208	17160	6.35	1820	286.5
-6.04	-.000196	53600	2.035	1710	841.5
-4.46	-.000158	90000	1.21	1380	1140.
-2.89	-.000137	33500	3.25	1197	368.
-1.42	-.000115	19600	5.56	1007	181.
.16	-.0000946	13760	7.94	829	104.4
1.63	-.0000785	10550	10.33	686	66.5
3.21	-.000062	8464	12.87	541.5	42.15
4.78	-.00005	7225	15.1	437	28.95
6.26	-.00004	6241	17.4	348	20.
9.4	-.000047	4942	22.0	409	18.6
12.43	-.000078	4186	26.0	681	26.2
15.49	-.0001368	3894	28.0	1192	42.6

Authority A. D.M. 1061.

EFFECT OF C.G. LOCATION ON VALUE OF K_{Mcg} AT ZERO LIFT :

(1)

C.G. at (40% M.A.C.
(45% below M.A.C.

Cellule α°	inches from C.G. to Vector	K_r	K_{Mcg}
-9.07	25.5	-.000615	-.000265
-6.04	90.0	-.000133	-.000202
-4.95	zero lift		
-4.46	63.7	.000132	-.000142
-2.89	15.0	.000387	-.000098
-1.42	5.4	.000631	-.0000576

K_{Mcg} at zero lift = -.000160

(2)

C.G. at (40% M.A.C.
(On M.A.C.

Cellule α°	inches from C.G. to Vector	K_r	K_{Mcg}
-9.07	26.1	-.000615	-.0002717
-6.04	94.5	-.000133	-.0002125
-4.46	72.2	.000132	-.000161
-2.89	18.5	.000387	-.000121
-1.42	7.6	.000631	-.0000811

K_{Mcg} at zero lift= -.000175

(3)

C.G. at (25% M.A.C.
(45% below M.A.C.

Cellule α°	inches from C.G. to Vector	K_r	K_{Mcg}
-9.07	16.7	-.000615	-.0001738
-6.04	81.3	-.000133	-.000183
-4.46	72.0	.000132	-.0001607
-2.89	23.6	.000387	-.0001543
-1.42	14.1	.000631	-.0001505

K_{Mcg} at zero lift = -.000170

EFFECT OF C.G. LOCATION ON VALUE OF $K_{M_{cg}}$ AT ZERO LIFT:

(4)

C.G. at (25% M.A.C.
(On M.A.C.

Cellule $\alpha°$	inches from C.G. to Vector	K_r	$K_{M_{cg}}$
-9.07	17.2	-.000615	-.000179
-6.04	85.7	-.000133	-.0001928
-4.46	80.7	.000132	-.000180
-2.89	27.4	.000387	-.000178
-1.42	16.6	.000631	-.000177

$K_{M_{cg}}$ at zero lift -.000185

EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD AT ZERO LIFT:

Normal tail load = $K_{M_{cg}} \times A_w \times \frac{C}{d} \times V^2$

(1) C.G.at (40% M.A.C.)
(25% below M.A.C.)

$d = 177.3 - (.40-.31) \times 59.15 = 177.3 - 5.3 = 172''$

Normal tail load = $K_{M_{cg}} \times 240.76 \times \frac{59.15}{172.0} \times V^2$

Cellule α°	$(82.8 \times V^2 \times K_{m_{cg}})$	= Normal Tail Load
-9.07	(82.8 x 17160 x -.000265)	= -376.2
-6.04	(82.8 x 53600 x -.000202)	= -896.0
-4.46	(82.8 x 90000 x -.000142)	= -1060.0
-2.89	(82.8 x 33500 x -.000098)	= -272.0
-1.42	(82.8 x 19600 x -.0000576)	= -93.5

(2) C.G. at (40% M.A.C.)
(on M.A.C.)

$d = 172''$

Normal Tail Load = $82.8 \times V^2 \times K_{M_{cg}}$

Cellule. α°		
-9.07	(82.8 x 17160 x -.0002717)	= -386
-6.04	(82.8 x 53600 x -.0002125)	= -945
-4.46	(82.8 x 90000 x -.000161)	= -1200
-2.89	(82.8 x 33500 x -.000121)	= -336
-1.42	(82.8 x 19600 x -.0000811)	= -131.6

EFFECT OF C.G. LOCATION ON VALUE OF NORMAL TAIL LOAD AT ZERO LIFT;

(3) C.G. at (25% M.A.C.)
(45% below M.A.C.)

$$d = 177.3 + (.31 - .25) \times 59.15 = 177.4''$$

$$\text{Normal tail load} = K_{M_{cg}} \times V^2 \times 240.76 \times \frac{59.15}{177.40}$$

$$= 80.3 \times K_{M_{cg}} \times V^2$$

Cellule α°	$(80.3 \times V^2 \times K_{M_{cg}})$	=	Normal Tail Load
-9.07	(80.3 x 17160 x -.0001738)	=	-241
-6.04	(80.3 x 53600 x -.000183)	=	-789
-4.46	(80.3 x 90000 x -.0001607)	=	-1160
-2.89	(80.3 x 33500 x -.0001543)	=	-415
-1.42	(80.3 x 19600 x -.0001505)	=	-237

(4) C.G. at (25% M.A.C.)
(on M.A.C.)

d= 177.4"

Cellule α°			
-9.07	(80.3 x 17160 x -.000179)	=	-246
-6.04	(80.3 x 53600 x -.0001928)	=	-830
-4.46	(80.3 x 90000 x -.000180)	=	-1303
-2.89	(80.3 x 33500 x -.000178)	=	-478.5
-1.42	(80.3 x 19600 x -.000177)	=	-278

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Design loads for
horizontal tail surfaces
for airplanes.

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